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GUIDANCE AND CONTROL OF THE MARINER PLANETARY SPACECRAFT

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GUIDANCE AND CONTROL OF THE MARINER PLANETARY SPACECRAFT

by John R. Scull[1]

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Abstract

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The unmanned exploration of the Moon and nearby planets imposes very significant requirements on the guidance and control system of both the booster rocket and the spacecraft. The booster rocket must be launched into a narrow, moving corridor with great precision within a limited period of time. After separation, the spacecraft must be capable of providing corrections to the trajectory and controlling the orientation of solar panels, antennas, and scientific instruments.

The guidance and control systems of the Mariner planetary spacecraft are described. Some of the techniques developed to provide redundant operation during the long interplanetary flight are discussed. Flight performance of the Mariner guidance and control systems is reported.

Author

I. Introduction

The challenge of planetary exploration requires continuous research

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[redacted]

and development to achieve mission goals. Larger, multi-stage launching vehicles provide more thrust; spacecraft and scientific instruments for measuring conditions in outer space have become more complex and have been further refined to achieve greater accuracy and reliability. Successful completion of these planetary investigations requires the solution of many difficult problems in guidance and control.

The permissible launching period for a planetary spacecraft is limited to approximately one month every two years. During this period a maximum of two hours per day is considered a suitable "window" for launching the spacecraft.

The spacecraft must be launched from a platform moving around the Sun at 66,000 miles per hour toward a destination planet moving 10,000 or 15,000 miles per hour faster or slower than the Earth and located hundreds of millions of miles away! It must pass near this target with a degree of accuracy much higher than that of the best military artillery; yet, it must have a very low probability of impacting on the destination planet to avoid contaminating the surface with an unsterilized spacecraft. This interception requires that the spacecraft pass within the confines of a narrow zone only a few thousand miles on a side to enable the scientific instruments to make measurements of the planet.

The components of the control systems must operate successfully over an extremely long lifetime in a very hostile environment. The length of the Venus trajectory, 180 million miles over a 109-day period, is short when compared to that of Mars, 325 million miles over an eight-month period.

Along the way, the orientation system must direct high-gain antennas back toward the Earth to communicate scientific and engineering measurements. The solar panels must be pointed toward the Sun in order to generate power required for spacecraft operation. Scientific instruments

must be properly oriented toward the planet during the encounter sequence to obtain measurements required for each experiment aboard the spacecraft. The accuracy of orientation must be relatively high to provide directional control of a mid-course rocket motor used for correcting the trajectory.

The foregoing accomplishments must be made while the spacecraft is traveling through outer space where it is exposed to vacuum conditions greater than those found in the best radio tubes, while being bombarded with charged particles, cosmic dust and meteorites. Due to the change in solar constant over the trajectory, the variation in radiant incoming heat is nearly two to one. The scientific payoff at the end of the mission is a scant half hour of data obtained during the time of closest approach to the planet. On some missions successful operation beyond this time is required in order to transmit data, recorded during planetary encounter, back to the Earth at a low rate.

II. Launch Vehicle Characteristics

The Mariner planetary spacecraft were launched on their voyages by the Atlas-Agena booster. The first-stage rocket, developed by Convair-Astronautics Division, was first flown in 1957. The booster is one which has been modified by the National Aeronautics and Space Administration to be used as a space launch vehicle. The Atlas is steered by gimballing the rocket engines, rather than by using conventional control fins. Attitude control is provided by a programmed autopilot overridden by radio commands from a radar system and computer located on the ground.

The upper stage of the booster rocket, the Agena, is produced by Lockheed Missiles and Space Division and was first flown in 1959. The Agena has been used as an upper stage on both the Thor and Atlas boosters.

It is steered by gimballed rocket nozzles and controlled during coast flight by small gas jet thrustors. Rocket motor thrust is terminated at the proper times on a signal derived from an on-board velocity meter. Attitude of the Agena stage is determined by gyroscopes and infrared horizon scanners.

The spacecraft and Agena stage are boosted into a parking orbit of approximately 100 nautical miles; the vehicle then coasts to the appropriate injection point. A coast period, ranging from a few minutes up to a maximum of about 20 minutes duration, is required. At injection, the Agena rocket motor is again fired to launch the spacecraft into its interplanetary trajectory. The accuracy of this type of guidance system for Venus and Mars missions is shown in Table I.

The Venus Mariner II spacecraft launched in 1962 was injected into a trajectory which would have passed 233,000 miles on the wrong side of Venus, while the Mariner IV Mars spacecraft was injected such that its miss distance was 158,679 miles from the planet.

III. Spacecraft Description

The Mariner II spacecraft launched toward Venus in 1962 weighed 447 lbs and was 11 ft, 11 in. high and 16 ft, 6 in. wide. Its mission was to conduct scientific measurements during the long, interplanetary flight and to pass within a zone 8,000 to 40,000 miles from the surface of Venus in such an orientation that the terminator could be scanned by a radiometer.

The power system consisted of two solar panels, totaling 27 square feet, and a rechargeable battery. Power consumed during the mission varied from 148 watts to 222 watts.

Scientific and engineering data were transmitted to receiving stations on Earth by an omnidirectional antenna and a high-gain steerable antenna. Commands from the ground were received through two command antennas and a command receiver. Orientation in three degrees of freedom was provided by the attitude control system. Cadmium-sulfide sensors directed the solar panels toward the Sun with an accuracy of one degree. The roll axis of the spacecraft was oriented toward the Earth by an Earth detector containing a photomultiplier and a vibrating-reed chopper. Actuating devices for the long cruising flight consisted of small 0.01 lb thrust gas jets fed from a source of pressurized nitrogen.

The spacecraft was tracked by the Deep Space Net (DSN) for several days after it was launched and deviations from the proper trajectory were determined at a central computing station. The velocity differential required to correct the trajectory, thus enabling the spacecraft to pass near the planet, was determined. Roll and pitch angles through which the spacecraft must turn and delta velocity required were transmitted to the spacecraft by radio command.

The mid-course correction system consisted of a 50 lb thrust hydrazine monopropellant rocket motor capable of imparting 200 ft per second velocity increment. During the thrusting period of the mid-course maneuver, the force from the gas jets was too small to correct for thrust misalignment torques. Small, postage-stamp size jet vanes, located in the rocket exhaust, were used to provide directional control.

The scientific instruments included in the Mariner II spacecraft consisted of a magnetometer, solar plasma probe, cosmic dust detector, charged particle and radiation detector, plus two radiometers for measuring planetary surface temperature at encounter.

The Mariner IV spacecraft launched toward Mars in 1964 weighed 575 lbs, stood 10 ft high, and was 22 ft, 7 in. wide. It was designed to carry scientific instruments for investigating the interplanetary region between Earth and Mars.

During planetary encounter a series of television pictures of the surface will be taken and recorded on magnetic tape. These pictures will be played back to the Earth at spaced intervals for six days immediately following closest approach. As the spacecraft passes behind the planet, the refraction and attenuation of the radio signal during occultation will be used to measure the properties of the Mars atmosphere.

The power for the spacecraft is provided by four solar panels, having a total area of 70 square feet, and a rechargeable storage battery. Engineering and scientific measurements are telemetered back to the Earth via redundant 10 watt radio transmitters. One transmitter is a cavity amplifier, while the other contains a traveling-wave tube.

Telemetry is transmitted over an omnidirectional antenna on the first half of the flight. In the latter portion of the trajectory

commands and telemetry are handled by a high gain directional antenna. Unlike the Mariner II design, the high-gain antenna of the Mariner-Mars spacecraft is not steerable, but is rigidly attached to the structure. This mechanization is possible due to the unique trajectory characteristics of the mission which allow a fixed-antenna orientation.

The attitude control system in the pitch and yaw axes is similar to that of the earlier Mariner, and solar orientation is sensed by cadmium-sulfide detectors. The reference for the roll attitude is the star Canopus, which is the second brightest star in the heavens and is located at nearly the south ecliptic pole. Canopus was selected as a roll reference, rather than the Earth, because Earth passes between the spacecraft and the Sun during part of the interplanetary trajectory. This causes the Earth to appear as a narrow crescent, as well as the Sun to appear in the field of view of the Earth sensor.

The Canopus star tracker uses an electrostatic image dissector as its detector, thus requiring no moving parts. Nitrogen gas jets and jet vane actuators, similar to those of the Mariner Venus design, are employed. Compensation for an unbalance in solar radiation pressure is provided by moveable paddles located on the tips of the solar panels. Each time a gas jet is actuated, the solar radiation pressure vanes are stepped differentially 1/100th of a degree, thereby compensating for any unbalance in external torques over a period of several days. An unbalance of 70 dyne-cm, caused by solar radiation pressure, biased the attitude control system on Mariner II to one side of the limit cycle. In the Mariner-Mars spacecraft these torques are reduced to below 10 dyne-cm, which reduces the number of times the gas jets are required to actuate. The accuracy of orientation in the Mariner-Mars spacecraft was improved

to 1/4 degree.

The midcourse correction employed by Mariner IV was somewhat like that of the Venus probe but contained an added provision for multiple restarts which allowed any additional vernier trajectory correction required later in the mission to be made.

IV. Sequence of Events

The flight sequence for the Mariner-Mars spacecraft began just prior to launch when the inhibit to the central timer was removed, permitting the timer to begin operating. Many of the components of the guidance and control system were turned on and operated during the boost period in order to telemeter their performance while in a high-acceleration environment.

Five minutes after lift off, when the booster reached an altitude where the atmospheric pressure was sufficiently low, the nose shroud was separated and the high-voltage power supplies of the transmitter and cruise science instruments were turned on.

Separation of the spacecraft and booster occurred after burnout of the Agena stage and about one minute later solar panels and solar vanes were deployed.

The attitude control system was activated and the Sun acquired approximately one hour after lift off.

During the next 15 hours, the spacecraft was rolled about the Sun line for calibration of the magnetometers. At the conclusion of this period, the Canopus sensor was turned on and a roll search initiated.

After Canopus was acquired the gyroscopes used for providing control during the search and acquisition modes were turned off.

The mid-course maneuver sequence was initiated one to 10 days following launch. Times for pitch- and yaw-commanded turns were transmitted from the Deep Space Net, along with motor burn duration. An additional ground command began the maneuver sequence. The gyroscopes were again turned on and allowed to warm up for about an hour, at which time the spacecraft was placed on inertial control and executed the predetermined angles in pitch and roll. The mid-course motor was ignited and burned for the prescribed time.

About an hour after ignition of the mid-course rocket motor the timer initiated an automatic reacquisition of the Sun and Canopus. Upon completion of this maneuver the logic inhibit of the solar vane system was removed and the vanes began their adaptive compensation for unbalanced solar radiation pressure.

Approximately one month after launch the data rate of the telemetry system was reduced from 33.3 bits to 8.3 bits per second.

The nominal angle for the field of view of the Canopus sensor was adjusted four times during the coasting trajectory. Canopus lies approximately 15 degrees away from the south ecliptic pole, thus in the trajectory around the Sun the angle between the Sun and Canopus may vary as much as 30 degrees.

Three months after launch the telemetry signal was switched from the omni-directional antenna to the fixed high-gain antenna.

Six hours prior to encounter the science instruments for the encounter sequence are activated.

The television system begins taking pictures 10 minutes prior to closest approach. All other encounter science instruments continue to operate for approximately six hours after the 22 television pictures

have been recorded on magnetic tape.

Thirteen hours after closest approach the tape recorder plays back the television pictures at a reduced data rate.

The mission concludes approximately 20 to 40 days after encounter when the combined effects of increased range and fixed antenna pointing error cause the communications system to exceed its threshold.

V. Reliability Enhancement

Extremely long operating times required of the Mariner spacecraft caused redesigns of the previous Ranger guidance and control system. Changes were made primarily to obtain longer life and redundant operation in the event of subsystem failure.

First to be eliminated for the cruise portion of the flight were the gyroscopes which provide rate damping to the attitude control system. The spin motor bearings of most gyroscopes have a life expectancy of from 1,000 to 2,000 hours, making them one of the lowest reliability components in the spacecraft. However, they are necessary for initial acquisition, controlled maneuvers, and inertial stabilization in case of Sun- or Canopus-sensor failure.

A synthetic rate-damping signal is generated by integration of the voltage applied to the solenoid valve of the gas jets. Since the actuation of the gas jets causes a fixed angular acceleration, the integral of this signal is proportional to the rate differential applied. This "derived rate" damping system was used on both Mariner II and Mariner IV. An attitude control system using derived rate damping is much less susceptible to noise which causes double gas jet pulses than one employing

gyro damping. Any disturbance in the spacecraft attitude resulting in Canopus leaving the field of view of the star sensor will automatically turn on the gyros and cause the system to go through a reacquisition sequence.

The solar radiation pressure control vanes added to the Mariner IV spacecraft are designed to reduce the unbalance torque acting on the pitch and yaw axes to nearly zero. In addition, the vanes produce a restoring torque which would cause the spacecraft to return to Sun orientation in the absence of gas-jet operation. The adaptive balance action of the control vanes is limited to $\pm$ 20 degrees from their nominal position to minimize the torque unbalance which would result if a vane actuator failed. The actuator and its associated electronics are designed to consume power only when actually in motion. The primary benefit from the solar vanes is a reduction in control jet gas consumption and resultant minimization in the number of times the solenoid valves must be actuated. This should markedly improve the reliability of the gas valves and reduce the probability of leakage caused by wear. The solar pressure balancing system is capable of compensating for an initial unbalance torque of up to 80 dyne-cm.

The total amount of pressurized nitrogen for the control jets is not reduced from that required without the balancing operation of the solar vanes. The amount of gas carried is sufficient to complete the mission, even if one solar vane should fail in either of its extreme positions. Solar vanes improve the overall reliability of Sun line attitude control by (1) reducing the requirements placed on the gas-jet system and (2) providing a redundant restoring torque in case of gas valve failures.

The gas system is further redundant in that each of the two valves providing a torque couple operates from a separate gas reservoir and regulator. If a gas valve fails in a closed position, the remaining valve providing torque around that axis will still continue to operate. Conversely, if a valve fails in an open position, the pressure from that tank would go to zero. Half the gas in the remaining tank would be dissipated in providing a torque to counter that from the leaking valve. Since the amount of gas carried in either tank is more than sufficient for the entire mission, the remaining gas in the second system would be capable of providing control for the duration of the mission.

Logic is built into the guidance and control and other systems to provide automatic mode switching should non-catastrophic failure occur. Automatic reacquisition of Canopus in the event of loss of track is described in a preceding section. Automatic switchover from one transmitter to a spare occurs if proper operation is not sensed at the end of a prescribed time period. The main power regulator is so designed that if the output voltage varies from the required limits, automatic switchover to an additional unit used during maneuvers will occur.

As an alternative to mode switching controlled by internal logic, the operating characteristics of the spacecraft may be changed by overriding ground commands. The spacecraft can be commanded to reject the star being tracked by the Canopus sensor and go through a roll search until a new star of the predetermined brightness falls within its gates. The center of the field-of-view of the Canopus tracker may be varied relative to the Sun by ground command, as well as by the internal sequencer. Internal logic, which causes the roll attitude

control system to go into an automatic roll search when certain Canopus brightness bounds are exceeded, may be removed by ground command. The spacecraft may be commanded to go into an inertially controlled mode if an attitude sensor fails, or to assume a new attitude by sending incremental angle commands, as well as holding a preset attitude.

Determination of system failures requires that a large proportion of the total telemetry channels be assigned for failure detection. Within the 20 attitude control channels and 22 channels for power system measurements, failure detection takes a higher priority than performance evaluation. Fault isolation down to the subsystem level may be determined on the ground to assist in making command override decisions.

Another major factor in reliability enhancement is spacecraft qualification testing which occurs at all levels, from overall spacecraft systems testing to that of initial component selection.

Many of the electronic components used to fabricate Mariner subsystems are subjected to rigid screening and aging processes prior to assembly. Component parts are selected from a small group of qualified types and go through further screening tests to eliminate units evidencing significant behavior variations. Component characteristics are measured before and after a temperature aging process and components showing variations beyond a predetermined level are rejected. Individual serial numbers are assigned to each electronic component part and an IBM record is kept showing its screening history. Failures occurring either at the component or subsystem level can be correlated with the performance of similar components taken from the same batch and selected by the same process. Thus systematic failures are detected and all components from a suspect batch are rejected.

Early in the program prototype units of all subsystems are assembled for a Proof Test Model (PTM). This is a complete spacecraft used to investigate any subsystem interface problems and evaluate overall system characteristics. Component or subsystem modifications determined from PTM evaluation may then be introduced into the flight spacecraft units during their fabrication process. Changes made to subsystem units subsequent to the beginning of PTM assembly must be checked out on this model before introducing modifications into flight units.

A Type Approval (T/A) unit is used to determine possible subsystem weaknesses prior to acceptance testing of the flight units. One unit, identical to each flight subsystem, is exposed to environments significantly above those expected during flight. Vibration levels 50 percent higher than those expected during the boost period are typical of type-approval testing. Other tests performed include high and low temperature testing in a vacuum, humidity, shock, static acceleration, rf interference, drop testing and handling.

Each flight subsystem must undergo Flight Acceptance (F/A) tests, similar to those described under Type Approval testing, but at environmental levels expected during actual flight, prior to delivery for assembly into the spacecraft.

After the flight spacecraft is assembled, subsystem and system functional tests are performed. When the spacecraft is considered completely flightworthy, it undergoes a series of mission tests which attempt to duplicate at the entire spacecraft system level a sequence of operations expected in actual flight. These include operation of all major elements of the spacecraft in a vacuum and simulated solar

radiation environment. The entire spacecraft must successfully pass a series of mission tests without a major failure before it is considered acceptable for launch.

A unit similar to the flight module is operated in a vacuum and variable temperature environment over a prolonged operating time, equal to that expected during the mission, in order to determine any time-dependent characteristics of the subsystems. From time to time during these life tests critical parameters are measured and variations from previous test results are determined.

Thus, it is seen that the reliability program necessary for long-duration missions is not only one of qualification testing, but, in reality, is a way of life requiring meticulous attention to detail throughout the entire program, from initial component selection through mission testing.

VI. Flight Test Results

The Mariner II spacecraft was launched on its way toward Venus at 1:53 AM, August 27, 1962. Shortly after separation Sun acquisition occurred in a normal manner. Seven days after launch, when the Earth was far enough away that its intensity was not too great for the dynamic range of the Earth sensor, the central computer and sequencer initiated Earth acquisition. After the spacecraft stopped its roll motion and the Earth was apparently acquired, the telemetered output of the brightness reading from the Earth sensor was considerably lower than it should have been if it were locked properly on the Earth. The actual intensity monitored was that which might be expected if the spacecraft were locked on the Moon. The strength of the radio signal transmitted from the high

gain directional antenna, however, indicated that the roll axis of the spacecraft was properly directed toward the Earth.

The mid-course maneuver was delayed several days from the time originally planned pending verification of the proper Earth lock. On September 4, 1962, it was established that the spacecraft was properly oriented on the Earth, not the Moon, and the midcourse sequence was commanded. The midcourse rocket motor ignited and burned for 27.8 seconds. Thrust was terminated when the velocity measured by an accelerometer was equal to the value commanded from the ground. The mid-course maneuver decreased the relative speed of the spacecraft with respect to the Earth by 47 feet per second and resulted in a correction of the Venus miss distance from 233,000 miles to 21,598 miles. The shaded area of Figure shows the acceptable aiming zone for the Mariner II mission. The final aiming point shown was the nominal point commanded by the mid-course maneuver. The actual point of flyby is shown as the predicted point on the figure. Although the dispersion of the midcourse maneuver was larger than intended, the point of closest approach fell well within the acceptable region for the mission. The Mariner II mid-course maneuver set a record for trajectory correction when the spacecraft was 1,492,000 miles away. This is the greatest distance from the Earth that such a maneuver has ever been successfully achieved.

After Sun and Earth reacquisition following mid-course, the brightness telemetry measurement from the Earth sensor was still considerably below the anticipated amount. As the intensity was approaching the threshold value, a transient, apparently caused by a meteorite hit, caused the logic of the attitude control system to turn on the gyros and initiate an Earth acquisition sequence. Before the gyros were fully up

to speed the Earth brightness jumped up to the proper value and remained at a normal level throughout the remainder of the mission. The exact cause for this anomaly has never been determined. It has been suggested that an internal reflection in the optical system of the Earth sensor was being sensed in the early part of the mission and, as the Earth brightness decreased due to the increasing distance from the Earth, this "flare" reached a low enough value that the proper object was tracked.

The temperature rise of the Mariner spacecraft as it decreased its distance from the Sun was much more rapid than predicted. By December 12, six of the temperature sensors exceeded their upper limits and the equipment was exposed to an environment far beyond the tested levels. Several of the subsystems, including the solar panels and the internal sequencer, suffered partial failures due to the high temperatures. The encounter sequence for scanning the planet was designed to be initiated by the internal sequencer, but three days prior to closest approach there was reason to believe that it would not function. A redundant sequence initiation provided by an Earth command was designed into the spacecraft. The command was transmitted from the Goldstone Tracking Station on the 109th day of travel and, 36 million miles away, the Mariner II responded and began its encounter sequence.

Twenty days after passing Venus the telemetered brightness of the Earth had reached its design limit. When the Woomera tracking station made a normal search for the spacecraft signal it could no longer be found. On succeeding days other DSN stations also searched in vain. Nevertheless, the Mariner II attitude stabilization system set another record, both the the longest continuous operation (129 days) and for traveling the greatest distance of any previous spacecraft (53.9 million

miles from the Earth).

During the 129 days of operation the gas jets of the attitude control system used only two of the four pounds of pressurized nitrogen gas. The limiting factors which terminated the mission were (1) temperature extremes beyond the design limits and (2) increasing Earth/spacecraft distances which caused the Earth sensor and communications to reach their capacity.

The Mariner IV spacecraft was launched toward Mars November 28, 1964. Sun acquisition was normal; however, some difficulty occurred in the acquisition of the roll reference star, Canopus. Since the absolute brightness of Canopus had not been measured from above the Earth's atmosphere, narrow intensity limits could not be used. In roll searching for Canopus, the spacecraft locked on several stars of almost comparable brightness located at the same angle away from the Sun. A provision was built into the system to allow an override and subsequent roll search initiation based on Earth command. Canopus was acquired, as indicated by the proper telemetered brightness of both Canopus and a simple Earth detector. It was then determined to proceed with the mid-course maneuver.

The maneuver sequence was commanded December 4, 1964; however, the initial starting transient of the gyroscopes caused the spacecraft to lose roll reference. Canopus had to be reacquired and the mid-course maneuver was postponed until the next day while the situation was analyzed. The second attempt at the mid-course maneuver was made November 5, 1964, and this time was completely successful. The commanded velocity increment of 51 feet per second reduced the miss distance of 158,679 miles to 5,536 miles from the surface. Figure shows the acceptable target region about Mars, the aiming point, and the closest approach points

determined from the first four trajectory computations.

Following the mid-course maneuver, the Sun and Canopus were reacquired. However, the Canopus tracker caused a loss of roll lock and automatic reacquisition was required several times. It was determined, from analysis of the telemetry records, that bright objects were coming within the field of view of the Canopus tracker and causing the intensity and error channels to initiate logic for a reacquisition sequence. One of these bright particles occurred shortly after a space science instrument telemetered a meteorite impact on the spacecraft.

It was concluded that the bright objects causing Canopus tracker difficulties were probably dust particles dislodged from the spacecraft following micrometeorite impact. An Earth command was sent to the spacecraft which removed the intensity and angular error logic of the roll control system so these bright particles would not cause automatic roll search. Since transmission of this command no inadvertent roll searches have occurred, although several bright particles have passed in and out of the field of view of the tracker.

The rate of gas consumption of the Mariner IV nitrogen gas supply is 0.03 pounds per day and the remaining life of the gas system is approximately four years.

Solar vanes have reduced the initial radiation pressure unbalance from 30 dyne-cm to below an average of 5 dyne-cm. On one day, December 15, 1964, the radiation pressure unbalance acting on the yaw axis of the spacecraft was so well balanced that a nine-hour interval occurred between successive firings of the gas jets. It has not been possible to completely balance the external torques acting on the spacecraft so that the gas jets are never actuated because the torques appear

to have a random value of ± 5 dyne-cm and reversals of torque have been observed over as short a period as one to two hours. This effect was not observed in the Mariner II spacecraft because of the presence of a high biasing torque.

V. Conclusions

The Mariner systems were specially designed to assure reliable operation of extremely long mission durations. These special considerations include:

1. Simplicity of design;
2. Reduction in the number of moving parts;
3. Automatic redundancy features which provide backup in the event of failure;
4. A large number of telemetry measurements to evaluate system and subsystem behavior;
5. Ground command;
6. Override capability for most backup functions;
7. A thorough testing and qualification program prior to sending the spacecraft on its long voyage into space.

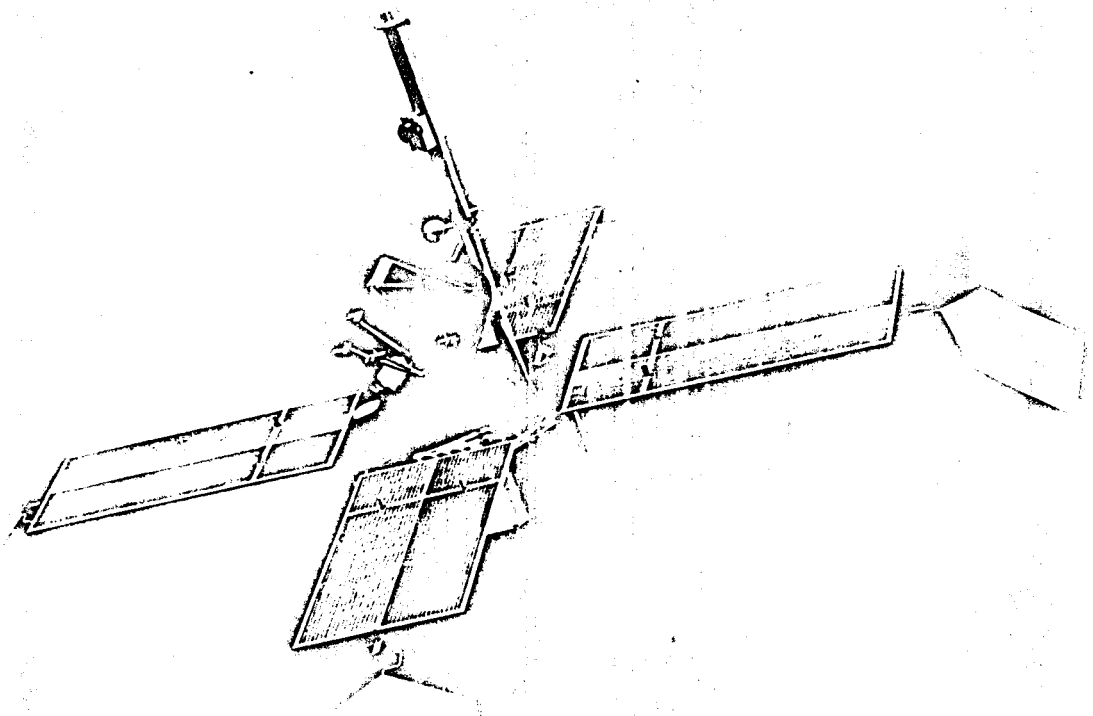
The resultant spacecraft, based on these considerations, have shown the ability to accomplish difficult mission tasks even in the presence of partial failures of components and in a hostile environment.

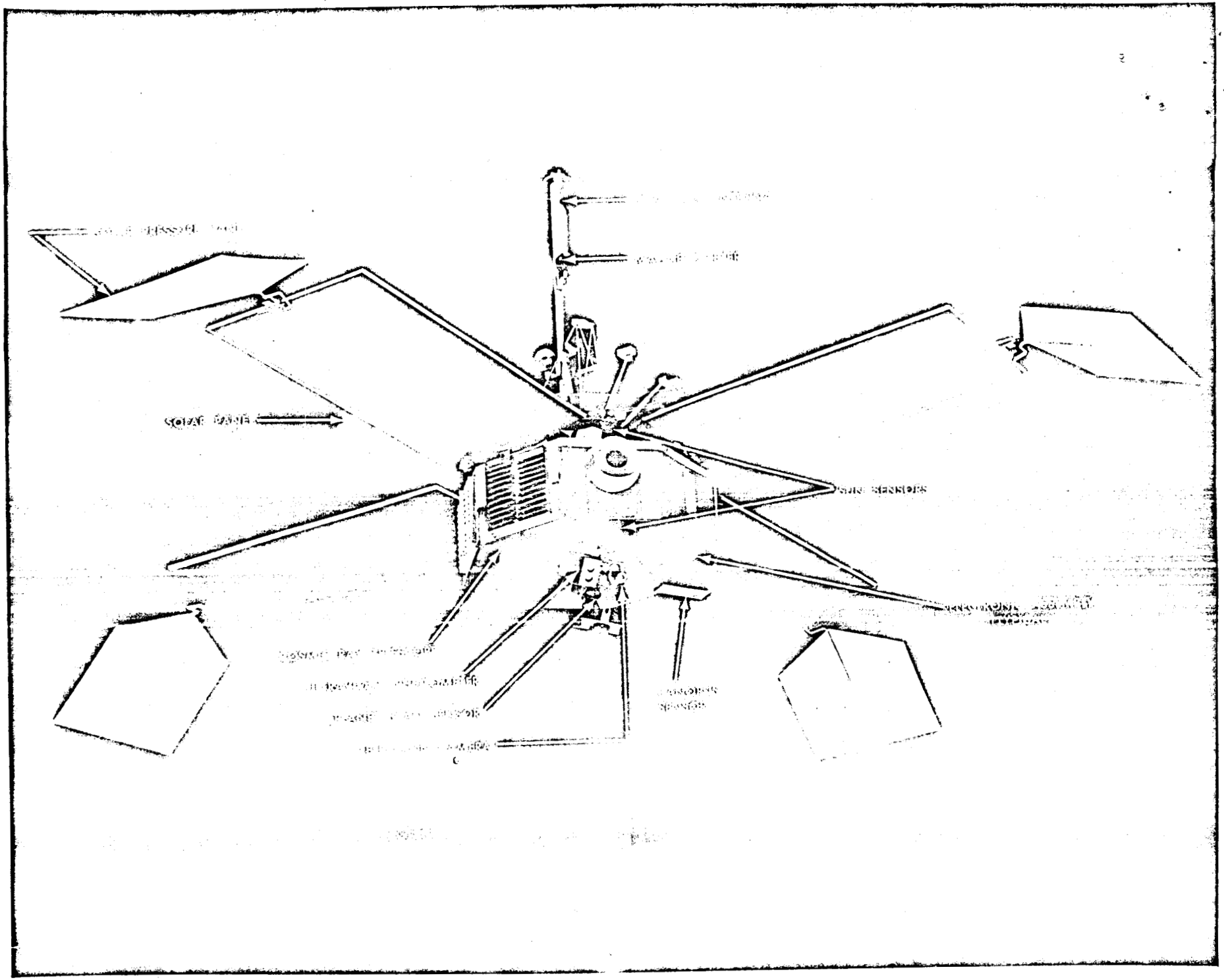
MIDCOURSE GUIDANCE PERFORMANCE

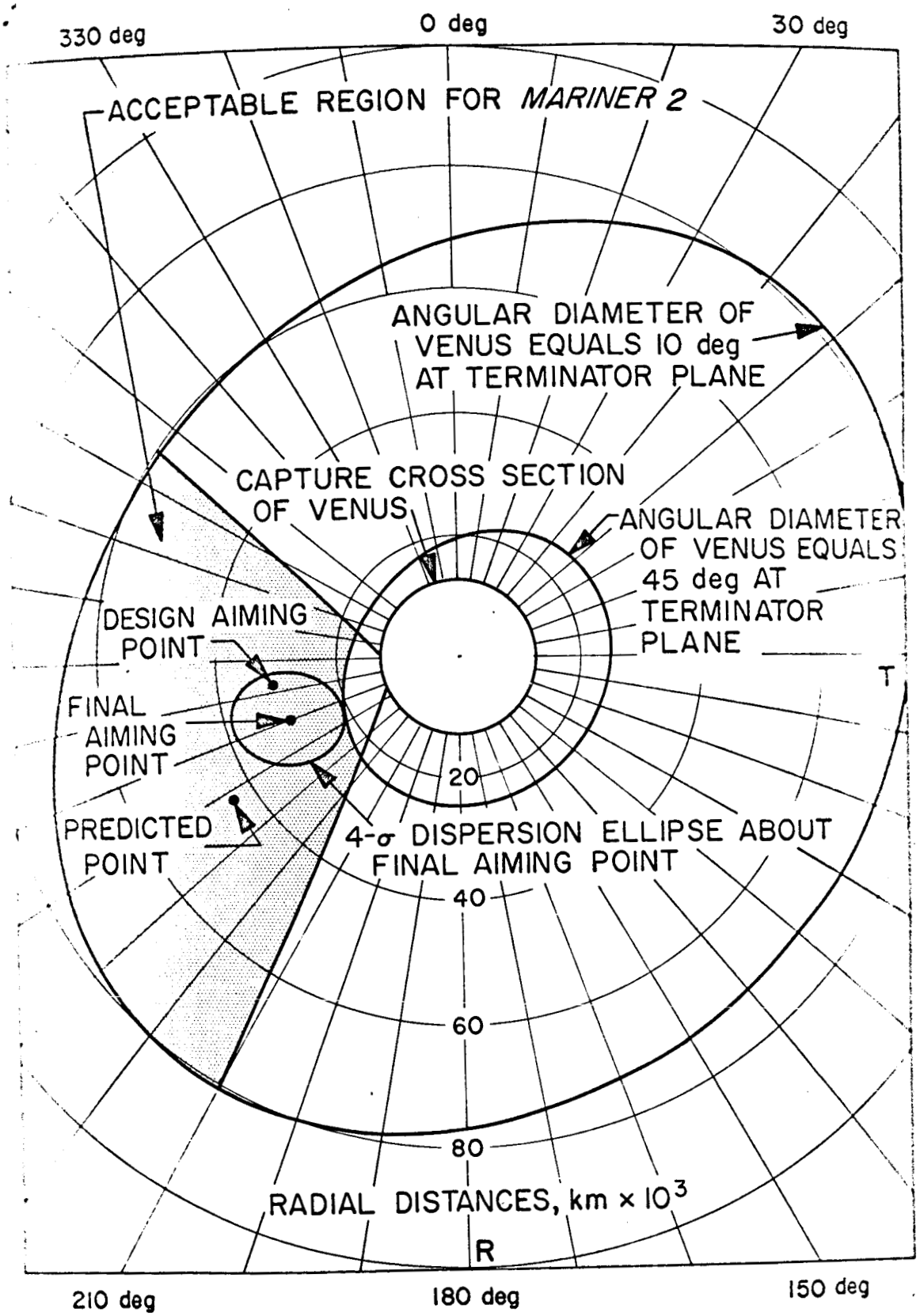
Target	(1) Miss due to representative injection guidance km	(2) Orbit determination accuracy from tracking accuracy km	(3) Midcourse maneuver to correct (1) m/sec	(4) Error due to maneuver km	(5) Total Accuracy (RMS of 2 and 4) km
Moon	6,000	10	40	64	65
Mars	500,000	2,500	20	5.400	6.000
Venus	300,000	1,000	20	2.700	2.900

Assumptions:

1. All quantities are one sigma
2. Tracking accuracy: 2×10^{-3} rad - 0.15 m/sec
3. Accuracy of maneuver: Pointing 0.5^{o} - Magnitude 1.0%






330 deg
0 deg
30 deg
ACCEPTABLE REGION FOR MARINER 2
ANGULAR DIAMETER OF VENUS EQUALS 10 deg AT TERMINATOR PLANE
CAPTURE CROSS SECTION OF VENUS
ANGULAR DIAMETER OF VENUS EQUALS 45 deg AT TERMINATOR PLANE
DESIGN AIMING POINT
FINAL AIMING POINT
PREDICTED POINT
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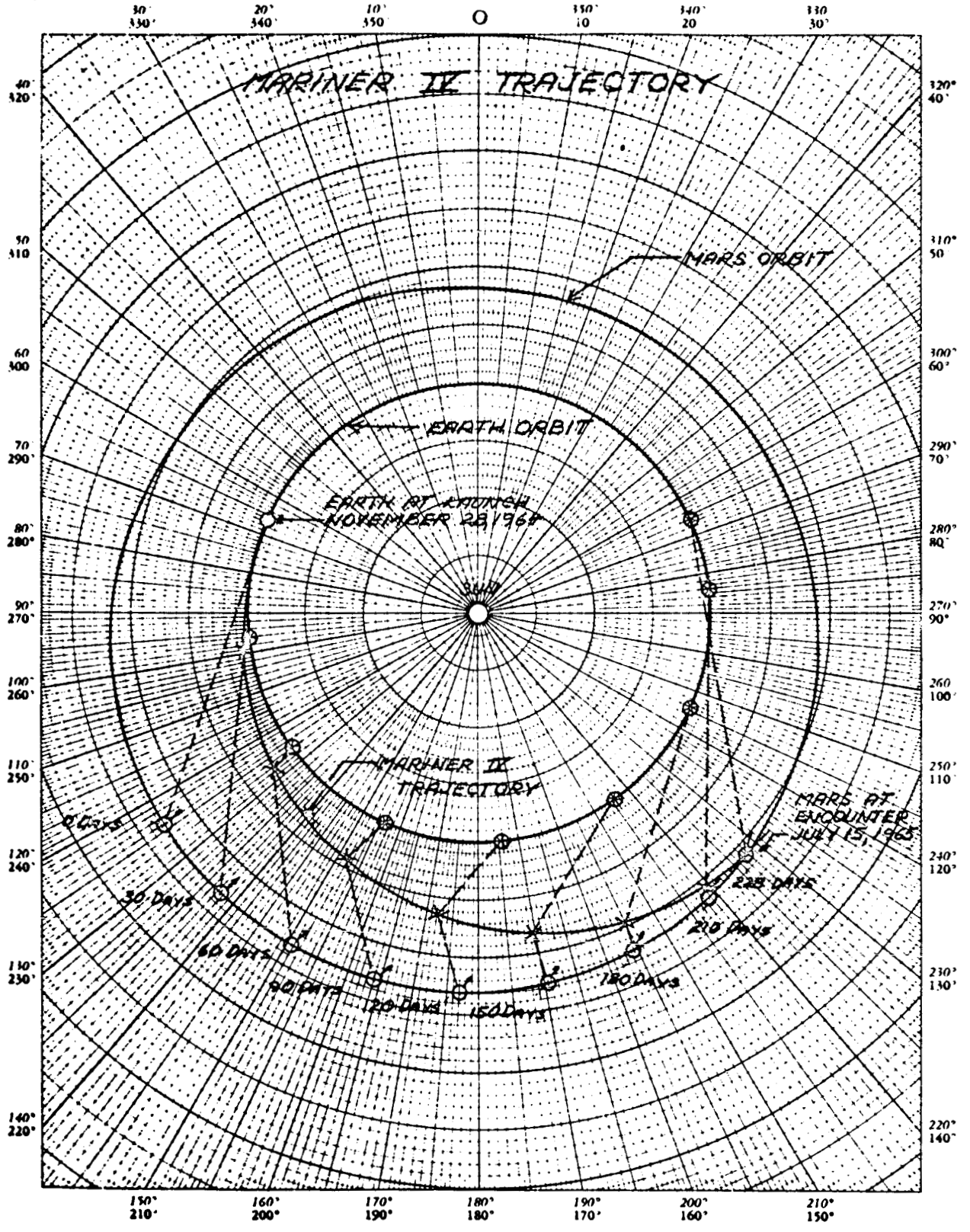
20
4-σ DISPERSION ELLIPSE ABOUT FINAL AIMING POINT
40
60
80
RADIAL DISTANCES, km × 10³
R
210 deg
180 deg
150 deg


MARINER IV TRAJECTORY
MARS ORBIT
EARTH ORBIT
EARTH AT LAUNCH
NOVEMBER 28, 1964
SUN
MARINER IV
TRAJECTORY
MARS AT
ENCOUNTER
JULY 15, 1965
0 Days
30 Days
60 Days
90 Days
120 Days
150 Days
180 Days
210 Days
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O
30° 330°
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10° 350°
350° 10
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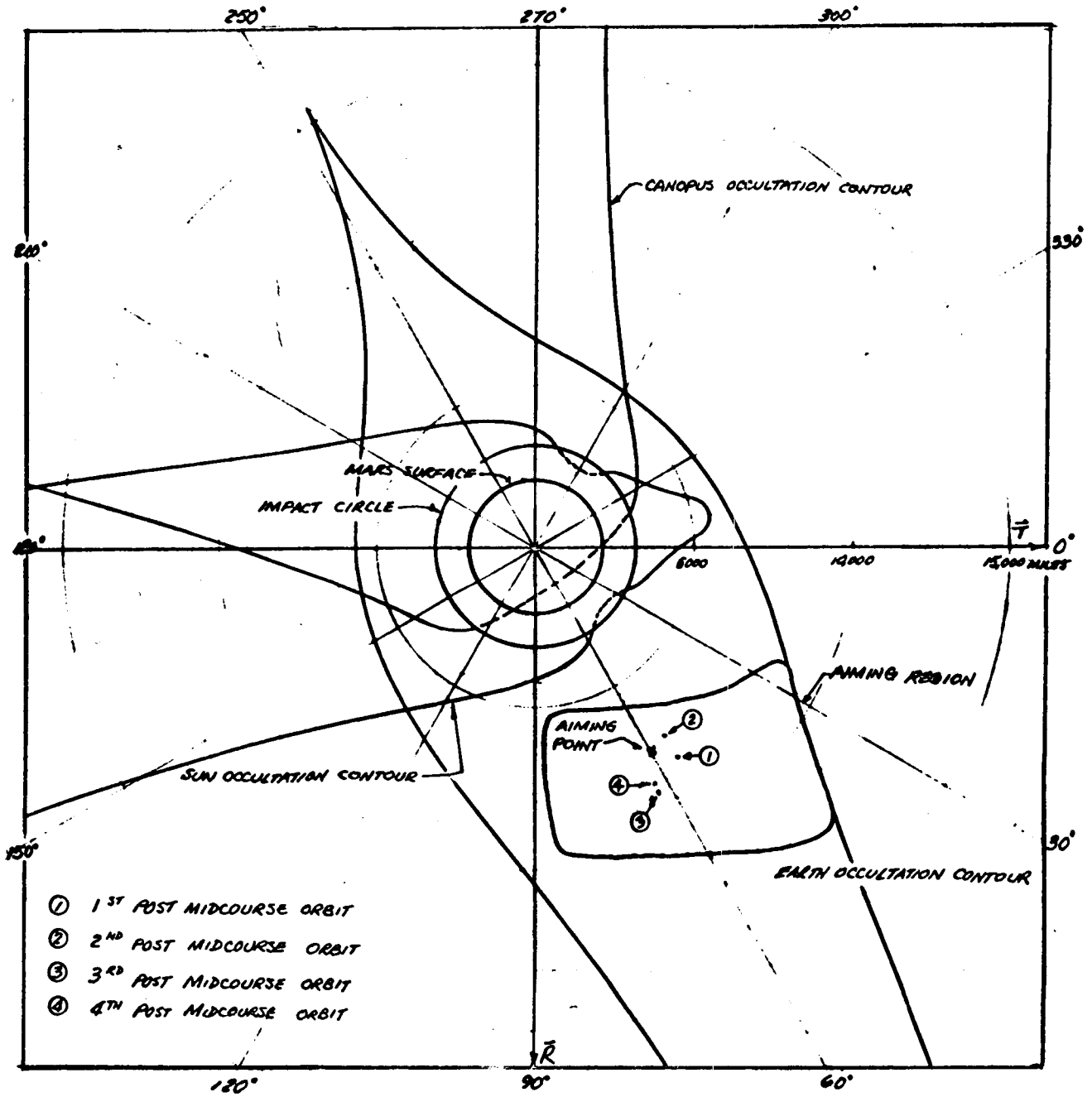
170° 190°
180° 180°
190° 170°
200° 160°
210° 150°

MARINER IV AIMING ZONE AND CONSTRAINT REGION



MIDCOURSE GUIDANCE PERFORMANCE

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Moon	6,000	10	40	64	65
Mars	500,000	2,500	20	5.400	6.000
Venus	300,000	1,000	20	2.700	2.900

Assumptions:

1. All quantities are one sigma
2. Tracking accuracy: 2×10^{-3} rad - 0.15 m/sec
3. Accuracy of maneuver: Pointing 0.5^{o} - Magnitude 1.0%